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CENTAUR/SURVEYOR NOSE FAIRING AERODYNAMIC HEATING INVESTIGATION

by C. W. Eastwood, Jr.

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

An experimental investigation was conducted to verify the structural integrity of the Centaur/Surveyor fiberglas nose fairing at the thermal and pressure environment predicted for the fourth Atlas/Centaur flight. Specimens of the fairing, including two coated with a protective subliming material, were exposed to the simulated aerodynamic heating and decreasing atmospheric pressure associated with the flight trajectory. Nose fairing specimen skin temperatures, bondline temperatures, and core pressures in the simulated environment are presented. Core pressure required to delaminate the skin-core assembly is given. Fairing skin temperatures from two subsequent Atlas/Centaur flights are compared with the experimental results.

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SUMMARY

The Centaur space vehicle utilizes a honeycomb cored, laminated fiberglas nose fairing to protect the payload from aerodynamic forces. In order to establish the structural integrity of the fairing at the thermal and pressure environment predicted for the fourth Atlas/Centaur (AC-4) flight, an experimental investigation was conducted at the Lewis Research Center. Specimens of the fairing were exposed to the simulated aerodynamic heating and decreasing atmospheric pressure associated with the flight trajectory, but not to aerodynamic forces or flight accelerations. Other samples of the fairing were coated with a thermostatic sublimate, which sublimes at 230° F (383 K), before exposure to the simulated environment in order to evaluate the temperature limiting effects.

Simulated aerodynamic heating raised the outer skin-to-core bonding agent temperature into the region of greatly reduced adhesive strength. Concurrently, as the environmental pressure was decreased to simulate ascent, the pressure of the entrapped core air increased from the thermal effects and resulted in a significant pressure differential that imposed forces on the weakened bonding agent. One of the uncoated specimens, which was exposed to maximum design heating, experienced catastrophic structural failure by sudden and complete delamination of the outer skin from the core. With test samples that were coated with the subliming material the outer bond line temperature remained below the specimen fabrication temperature and no significant increase of core pressure was noted. By application of a sufficient thickness of 230 sublimate over the outer skin, the ability of the fairing structure to withstand the thermal and atmospheric environment was verified.

INTRODUCTION

The Centaur/Surveyor nose cone serves as a low drag fairing to minimize the aerodynamic loads and aerodynamic heating on the vehicle, the forward mounted equipment, and the payload while the vehicle is ascending through the atmosphere. It is a lightweight clam-shell type conical shroud mounted on the forward end of the Atlas/Centaur vehicle. To accomplish its function, the fairing must withstand the flight environment from launch through separation and jettison. While protecting the spacecraft, the nose fairing must withstand severe environmental conditions. The design and construction must therefore provide full protection under the conditions encountered with the lightest weight structure possible.

Basically, the conical nose fairing is a lightweight shell of phenolic glass fiber sandwich consisting of a honeycomb core with fabric laminates bonded to the outer and inner core faces. The strength of the assembly is limited by the bonding agent. The load carrying capability of the bonding agent decreases with increasing temperature. As the velocity of the vehicle increases during ascent through the atmosphere, the temperature of the nose fairing rises due to aerodynamic heating. Maximum temperatures occur when ambient pressures are near zero. At the same time the loads on the bonding agent are significant because of the positive air pressure trapped in the honeycomb core cells and the low ambient pressure.

An aerodynamic heating analysis of the nose fairing for the fourth Atlas/Centaur flight (AC-4 vehicle) indicated that maximum surface temperatures would occur on the nose cap (fig. 1) and on the conical section. The cap, being an uncored laminate, would experience no cell pressure loading. However, the cored conical section would be subjected to the pressure loads. From the analysis it was determined that the maximum outer bond line temperatures of the cone would be from 765° F (680 K) to 925° F (769 K) depending upon the location on the conical section. The adhesive strength of the bonding agent markedly decreases at these temperatures. In addition, core pressures were not amenable to accurate calculations and thus, loading on the bond line was uncertain. As a result, application of a subliming heat absorber on the outer surface of the nose fairing was investigated. The sublimate used was Thermolog T-230 (230 sublimate) manufactured by Emerson Electric Company, St. Louis, Mo.

A development test program was initiated to (1) determine the core pressure history, (2) confirm the analytical values of bond line temperatures, and (3) verify the structural integrity of the conical fairing when surfaced with 230 sublimate as a heat absorber.

NOSE FAIRING CONFIGURATION

Nose Fairing Description

Principal shell components of the fairing are a truncated cone, a cylindrical barrel attached at the cone base, and a hemispherical cap at the apex (fig. 1). The fairing assembly is composed of two halves with the split line joined by pyrotechnically actuated latches for jettisoning. Attachment to the forward end of the Centaur propellant tank is accomplished by a circumferential band. The band is pyrotechnically severed before jettisoning. Hinges at the interface guide the rotation of the shroud halves during the beginning of separation. Vent ports are spaced around the barrel section and relieve the atmospheric pressure in the payload and equipment compartments as the vehicle ascends. Fairing separation and jettison occurs at a time during flight after aerodynamic loads and heat flux have become negligible. Gas discharge thruster bottles located in the front end of the conical section supply the separation forces.

The hemispherical cap is a shell fabricated from phenolic glass fiber laminates and covered with a high silica glass layer. The conical and barrel section are of sandwich-type construction. Glass-fabric-reinforced plastic honeycomb constitutes the core. Skins composed of phenolic resin glass cloth are bonded to the core with epoxy phenolic adhesive. Outer surfaces of the assembly are treated with an epoxy sealer and epoxy enamel. Skin and core thickness vary with vehicle station as shown in figure 1.

Test Specimen

Six test panels were used in the test program as fairing specimens. Four of these panels, each 24 inches (60.96 cm) square, were obtained from the cone section of a flight-type fairing and were designated as specimens 1 to 4. The other two panels were 7 inches (17.78 cm) square and were fabricated to the same specifications as the flight fairing. The two smaller panels were coated on the outer skin with 0.04 inch (0.1 cm) of the 230 sublimate.

The exposed core edges of specimens 1 to 4 were covered with epoxy glass cloth laminates extending 1 inch (2.54 cm) onto the skin faces. The core edges and inner skin of specimens 5 and 6 were undercut 1/2 inch (1.27 cm) all around and sealed with laminates of epoxy glass cloth.

Instrumentation of these panels consisted of iron constantan thermocouples and strain-gage type pressure transducers. The types of instrumentation installation are presented in figure 2. All thermocouples were located within a 1/2-inch (1.27-cm) radius of the center of the panels. Each of the specimens had a pressure tap mounted on the inner skin, except for specimen 1, which had no pressure sensor. The pressure tap was located 3 inches (7.62 cm) off center on specimens 2 to 4, on center on specimen 5, and $1\frac{1}{2}$ inches (3.81 cm) off center on specimen 6. The number of thermocouples and pressure sensors for each specimen and the type of installation are listed in table I.

Specimen 4 had skin-to-core delaminations formed prior to test by locally crushing the core in specific areas. Purpose of the delaminations was to simulate possible damage to the fairing during shipping and handling. The core was crushed by pressing a wooden sphere of 8-inch (20.3-cm) radius on the panel skin at the desired location and crushing the core immediately beneath without doing damage to the skin itself. This action separated the core and skin at the bond line. Two 2-inch (5.08-cm) diameter flaws were formed on the panel; one centered under the outer skin and one 3 inches (7.62 cm) off center under the inner skin.

Specimens selected for specific investigation are as follows:

Specimens					Skin thickness				230 sublimate coating		Effects investigated
Number	Size		Thickness		Outer		Inner				
	in.	cm	in.	cm	in.	cm	in.	cm	in.	cm	
1	24 by 24	60.9 by 60.9	$1\frac{1}{4}$	3.17	0.04	0.1	0.04	0.1	None		Maximum design heating
2	24 by 24	60.9 by 60.9	$1\frac{1}{4}$	3.17	0.04	0.1	0.04	0.1	None		Maximum design heating
3	24 by 24	60.9 by 60.9	$1\frac{1}{4}$	3.17	0.04	0.1	0.04	0.1	None		3-sigma low-trajectory heating; post-test skin porosity
4	24 by 24	60.9 by 60.9	$1\frac{1}{4}$	3.17	0.04	0.1	0.04	0.1	None		Protective coating sublimation temperature on bond line and core pressure; fairing structural margin for maximum core pressure
5	7 by 7	17.8 by 17.8	$1\frac{1}{2}$	3.8	0.04	0.1	0.04	0.1	0.04	0.1	230 sublimate coating for 3-sigma low-trajectory heating; structural integrity of fairing
6	7 by 7	17.8 by 17.8	$1\frac{1}{2}$	3.8	0.04	0.1	0.04	0.1	0.04	0.1	230 sublimate coating for 3-sigma low-trajectory heating; structural integrity of fairing

TEST FACILITY

Equipment and Instrumentation

Testing was performed in the Electric Propulsion Laboratory of NASA Lewis Research Center. The vacuum tank used to simulate the pressure environment was cylindrical with an inside diameter of 4 feet (1.22 m) and a depth of 5 feet (1.52 m). The tank was connected to a large capacity pumping system capable of duplicating the pressure profile predicted for a maximum heating trajectory flight from lift-off plus 50 seconds (T + 50 sec) through nose fairing jettison at T + 200 seconds.

Aerodynamic heating was simulated by radiant lamps mounted in gold plated reflectors. Two lamp banks were used; a large bank for heating the 24-inch- (60.9-cm-) square specimens (specimens 1 to 4) and a smaller bank for the 7-inch- (17.8-cm-) square specimens (specimens 5 and 6). The lamp banks were connected to a variable voltage source for regulating input power that controlled the heating rate to the test specimens.

The system used to pressurize specimen 4 consisted of a cylinder of pressurized nitrogen gas, control valves, and a supply line. A sectional view of the vacuum tank, a lamp bank, and the pressurizing system is shown in figure 3.

Vacuum tank pressures and specimen core pressures were sensed with strain-gage-diaphragm type transducers. Temperatures were sensed with iron-constantan thermocouples. Lamp bank voltage and current were measured and indicated on standard meters. All measurements were recorded on an oscillograph using light beam galvanometers. Temperatures from the first three tests had accuracies estimated to be ±10 percent of indicated reading ($^{\circ}$F) due to thermocouple installation difficulties. Improved methods of installation on the subsequent four tests yielded temperature readings with accuracies estimated to be ±5 percent of indicated reading ($^{\circ}$F).

Control and Calibration

Evacuation of the test tank was regulated by manual control of the facility pumping system. A comparison of the test tank pressure and the predicted ambient flight pressure is shown in figure 4. Good simulation was not obtained for the first 50 seconds due to limitations in the facility controls. This variation was not considered detrimental since the predicted aerodynamic heating was not significant until T + 60 seconds.

Heating for test 1 was regulated by monitoring specimen outer surface temperature against a predicted flight temperature-time curve and manually controlling voltage input to the lamp bank. For the remaining tests, heating was regulated by manually controlling the voltage input to duplicate voltages calibrated to produce the flight heat flux.

The latter method of regulation necessitated calibration of the lamp bank input voltage against heat flux to the specimen. This was done in the vacuum tank using a slug calorimeter mounted on a sample specimen panel.

TEST PROCEDURE

The specimen assigned to a particular test was clamped in the support fixture and mounted in the vacuum chamber as shown in figure 3. The recorders were turned on 5 seconds (T - 5) before time zero. Evacuation of the tank was begun at T - 0 second, which corresponds to lift-off for an actual flight, and continued at the flight simulation rate. The heating was started at the programmed time and was applied according to the specified profile. At approximately T + 220 seconds, the test was terminated.

Test 1 heating was controlled to follow the design maximum outer skin temperature program. All other test specimens were heated using the following predicted AC-4 flight heat flux profiles at the specimen outer surface:

Test 2 - at maximum design heating

Tests 3, 5, and 6 - at 3-sigma low trajectory

Tests 4 and 4A - at 3-sigma low trajectory with reduced outer skin heating simulating the effect of a 0.04-inch (0.1-cm) coating thickness of 230 sublimate surface material

A 3-sigma low trajectory is the minimum altitude trajectory predicted by a 3-sigma (99.87 percent) probability range resulting from all influencing dispersions about the normal trajectory. The minimum altitude trajectory heat flux profile was selected because it results in maximum heating for the nose fairing surface within the 3-sigma trajectory range. Heat flux profile values are listed in table II.

Specimen 3 was removed from the test tank subsequent to test 3 and was pressurized in the core area of the sample through the pressure sensing port with gaseous nitrogen to a differential pressure of 7 psi (4.82 N/cm^2). In test 4A, specimen 4 was retested and then subjected to a core pressure buildup to determine delamination structural margin. At T + 165 seconds, while the specimen was in the chamber at essentially zero ambient pressure, gaseous nitrogen was fed into the test panel through the pressure tap. The core pressure was raised in three increments of approximately 15 psi (10.3 N/cm^2) each.

RESULTS AND DISCUSSION

Experimental Data

Unprotected specimens of the flight fairing (specimens 1 and 2) failed when subjected to maximum design heat flux. The capability of a subliming surface material to limit the specimen temperatures to structurally safe values and prevent failure was confirmed. The test results are discussed in the following paragraphs and are summarized in table III.

In test 1, which represented maximum design heating, the outer skin maximum temperature was approximately 3 percent higher than had been planned and occurred at T + 143 seconds. At that time, the inner skin temperature had increased to 93° F (307 K) from its initial value of 80° F (300 K). It continued to rise to 125° F (325 K) at test termination. Curves of these parameters are shown in figure 5(a). Outer bond line temperature and core pressure was not measured in this test. The surface layer separated from the other layers of the laminated outer skin in the central surface area of the specimen and appeared to be void of all epoxy. Apparently the epoxy "boiled off" in the pressure-thermal environment.

Maximum outer skin temperature in test 2, which was conducted at maximum design heating, occurred at T + 147 seconds and was approximately 7 percent, 55° F (286 K) greater than the analytically predicted value. Bond line temperature was much lower than expected and was not considered valid because the thermocouple installation did not permit good contact with the skin surface. Inner skin temperature at that time was 28° F (15.5 K) higher than the initial value of 80° F (300 K). The core pressure commenced to decrease after the start of the test and reached a value of 13.5 psia (9.3 N/cm^2) at T + 84 seconds. It then increased to a maximum of 17.2 psia (11.85 N/cm^2) at T + 120 seconds. Subsequently, it dipped and finally increased to 15.5 psia (10.7 N/cm^2) at T + 147 seconds at which time the outer skin completely delaminated from the core. The temperature and pressure measurements are shown in figure 5(b).

The data from test 3 using 3-sigma low trajectory heating is shown in figure 6. Maximum outer skin temperature was lower, as expected, than in tests 1 and 2. Core pressure increased to a maximum of 17.0 psia (11.7 N/cm^2) at T + 140 seconds and then rapidly dropped off to 3.8 psia (2.62 N/cm^2) at T + 220 seconds. No delaminations or failures were evident. A porosity check of the outer skin was made following test 3 with the specimen removed from the vacuum chamber. The surface paint, sealer, and epoxy appeared to be boiled away from exposure to the test environment. However, when the pressure was increased in the central core area, 7 psi (4.82 N/cm^2) were require to force the gaseous nitrogen through the skin. Cooling of the specimen to room temperature during this check permitted solidification of the remaining epoxy in the outer laminate and was considered responsible for the pressure increase necessary to demonstrate porosity.

Tests 4 and 4A using 3-sigma low trajectory with reduced outer skin heating simulating the effect of a coating of 0.04-inch (0.1-cm) thickness of 230 sublimate produced considerably lower skin and bond line temperatures. Temperatures and pressures obtained from these tests are presented in figures 7(a) and (b). Core pressure in test 4 decreased slightly until T + 100 seconds, then increased to 14 psia (9.65 N/cm^2) at T + 130 seconds. The pressure remained at that value until T + 200 seconds when it started decreasing. In test 4A (using the same specimen as in test 4) core pressure decreased steadily from the start of test until, at T + 160 seconds, it was 13.8 psia (9.52 N/cm^2). At that time the specimen was pressurized from an external source of gaseous nitrogen. The difference in core pressures between test 4 and 4A prior to the nitrogen pressurization, probably was due to some "boiloff" of epoxy from the specimen during test 4 making it more porous during test 4A.

Core pressure, which was built up with nitrogen gas in test 4A, was started at T + 160 seconds and was continued until T + 200 seconds when 50 psia (34.4 N/cm^2) were reached as shown in figure 7(b). It was held at this value until at T + 223 seconds the inner skin completely separated from the core. Since the specimen was taken from an actual fairing, it had the associated fairing curvature. Therefore, the internal pressure subjected the inner skin to buckling and imposed a peeling force on the inner bond line. It is probable that delamination started at the point of the fabricated flaw where the pressure load per inch of bond line would be greatest. Since the 2-inch (5.08-cm) diameter delamination would cause the pressure load per inch of bond line to be greater than that from detectable manufacturing flaws, this test is considered conservative. Further, since the failure pressure was three times that which would be achieved in the normal trajectory, the structural margin would be greater than 3.

Tests 5 and 6, performed with 3-sigma low trajectory heating, were made with 0.04 inch (0.1 cm) of 230 sublimate applied to the outer skin surface of the specimens. Test 5 was not valid. Excessive heating was imposed on the specimen when control of voltage to the heating lamps was lost early in the run. However, the outer skin and bond line temperatures did not exceed 230° F (383 K) until the 230 sublimate completely sublimed at T + 130 seconds. Subsequently, temperatures and core pressure increased until the end of the test run. These measurements are shown in figure 8(a).

Test 6 temperatures of outer skin and bond line reached 200° F (367 K) and 170° F (350 K), respectively, at T + 130 seconds. Then, they increased slowly to a maximum of 235° F (386 K) and 220° F (378 K) at T + 180 seconds. Core pressure decreased to 13.8 psia (9.52 N/cm^2) at T + 130 seconds and then increased to 14 psia (9.65 N/cm^2) at T + 200 seconds. Thereafter, pressure decreased as shown in figure 8(b). Post test measurement of the remaining subliming material thickness on specimen 6 indicated a loss of approximately 0.01 inch (0.025 cm) (25 percent) during the test.

Comparison with Flight Data

AC-4 flight. - The fourth Atlas/Centaur flight trajectory (ref. 1) during booster phase was above normal but within 3-sigma value. Temperatures under the first layer of fiberglas of the outer skin were measured with thermocouples installed in a manner similar to those on test specimens 5 and 6. Also, similar to specimens 5 and 6, the fairing was coated with a 0.04-inch (0.1-cm) thickness of 230 sublimate.

Outer skin temperatures at station 125 on the conical section of the fairing are shown in figure 9(a). The shape of the outer skin temperature-time curve is similar to that from test 6 (fig. 8(b)). Maximum temperature indicated was 145° F (336 K) compared with 235° F (386 K) in test 6. Overall accuracy of the flight instrument system was estimated to be ±5 percent of range, 75° to 1115° F (42 to 619 K). This amounted to $\pm 52^\circ$ F (29 K) and established a value of 178° F (354 K) as the lower edge of the accuracy band for the predicted skin temperature of 230° F (383 K), the sublimation temperature of 230 sublimate.

AC-6 flight. - The flight trajectory of the sixth Atlas/Centaur vehicle (ref. 2) during the atmospheric portion of the flight was normal. The fairing had 0.027 inch (0.068 cm) of 230 sublimate on the surface and thermocouples installed under the first fiberglas layer of the outer skin.

Temperatures of the outer skin at station 72 reached a maximum value of 188° F (360 K) compared with 235° F (386 K) indicated in test 6. The flight system end-to-end accuracy was estimated to be ±5 percent; the same as for flight AC-4. Therefore, the maximum indicated temperature of 188° F (360 K) was within the accuracy band for the expected temperature of 230° F (383 K). The outer skin temperature-time data for AC-6 flight is shown in figure 9(b). The temperature-time profile is similar to that from AC-4 flight and to the profile from test 6 (fig. 8(b)).

CONCLUSIONS

The Centaur/Surveyor nose fairing aerodynamic heating test results show that during vehicle ascent through the atmosphere, the air trapped in the nose fairing honeycomb core will increase in pressure from the aerodynamic heating as a function of the trajectory. Bond line temperature also will be dependent on the flight path.

For maximum design aerodynamic heat flux, temperatures developed at the outer skin bond line and pressures in the honeycomb core will seriously weaken or fail the fairing structure. Three-sigma low trajectory heating will raise the outer skin and bond line temperatures to the point where epoxy boiloff will permit core pressures to bleed down through the skin, but will reduce the fairing strength in the area of the loss of epoxy. Tests showed that for heating from a 3-sigma low trajectory an adequate surface coating of 230 sublimate will maintain the fairing temperatures and core pressures well below critical values for structural integrity of the fairing.

A 230 sublimate coated nose fairing with a 2-inch (5.08-cm) diameter skin-to-core delamination type of flaw will be capable of successfully withstanding the aerodynamic heating and core pressure experienced in a 3-sigma low flight trajectory. The ultimate strength of the bond line will be greater by a factor of approximately 3 than the core pressure loading developed under these conditions.

The structural integrity of the fairing was further verified by AC-4 and AC-6 flight results.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 13, 1968,
491-05-00-02-22.

REFERENCES

1. Staff of Lewis Research Center: Postflight Evaluation of Atlas-Centaur AC-4 (Launched December 11, 1964). NASA TM X-1108, 1965.
2. Staff of Lewis Research Center: Postflight Evaluation of Atlas-Centaur AC-6 (Launched August 11, 1965). NASA TM X-1280, 1966.

TABLE I. - THERMOCOUPLES AND PRESSURE SENSORS

Specimen	Transducer type	Type of installation (a)	Number of transducers
1	Thermocouples	A	2
		G	1
2	Thermocouples	A	3
		C	2
		B	1
		G	1
	Pressure sensor	J	1
3	Thermocouples	A	2
		C	2
		D	1
		G	1
	Pressure sensor	J	1
4	Thermocouples	A	1
		B	1
		E	2
		F	2
	Pressure sensor	J	1
5	Thermocouples	H	2
		I	2
	Pressure sensor	K	1
6	Thermocouples	H	2
		I	2
	Pressure sensor	K	1

[a]Type of installation: A, thermocouple and leads on outer skin surface; B, thermocouple at outer skin bond line with taut leads through specimen; C, thermocouple on outer skin with taut leads through specimen; D, thermocouple at outer skin bond line with leads on outer surface; E, thermocouple on outer skin and slack leads through specimen; F, thermocouple at outer skin bond line with slack leads through specimen; G, thermocouple and leads on inner skin surface; H, thermocouple and leads under first layer of outer skin; I, thermocouple and leads in outer skin bond line; J, pressure sensor installed 3 in. (7.62 cm) off center through inner skin; K, pressure sensor installed $1\frac{1}{2}$ in. (3.81 cm) off center through inner skin. (See fig. 2 for drawing of installation methods.)

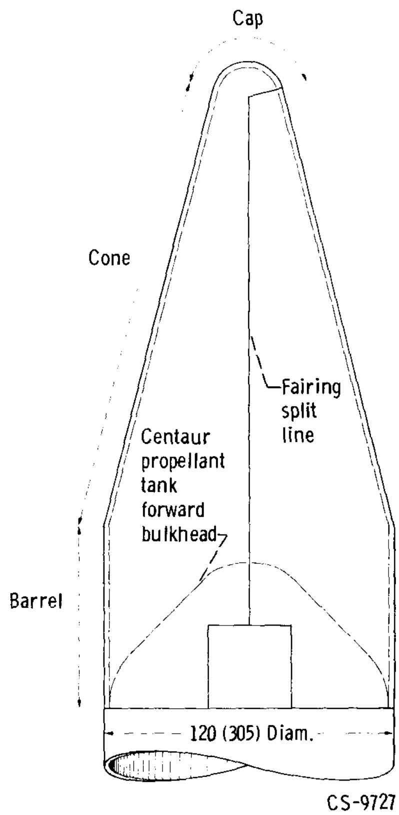
TABLE II. - HEAT-FLUX VALUES

Time, sec	Maximum design heating				3-sigma low-trajectory heat flux		3-sigma low-trajectory (with effect of 230 sublimate on fairing surface) heat flux	
	Heat flux		Outer skin temperature					
	Btu/(ft^2)(hr)	W/m^2	^{o}F	K	Btu/(ft^2)(hr)	W/m^2	Btu/(ft^2)(hr)	W/m^2
60	500	1 576	80	300	----	------	----	------
80	1977	6 240	130	328	646	2 038	578	1 820
90	4311	13 580	207	370	2300	7 260	----	------
100	6350	20 020	320	438	4540	14 300	3860	12 180
110	7330	23 060	435	497	5550	17 480	----	------
120	7910	24 920	552	561	6260	19 720	4960	15 630
130	7860	24 760	658	620	6620	20 840	----	------
140	7070	22 280	748	670	6280	19 780	4480	14 100
150	4460	14 050	790	694	5310	16 720	----	------
160	3570	11 250	812	706	2750	8 670	1430	4 510

TABLE III. - SUMMARY OF TEST CONDITIONS AND RESULTS

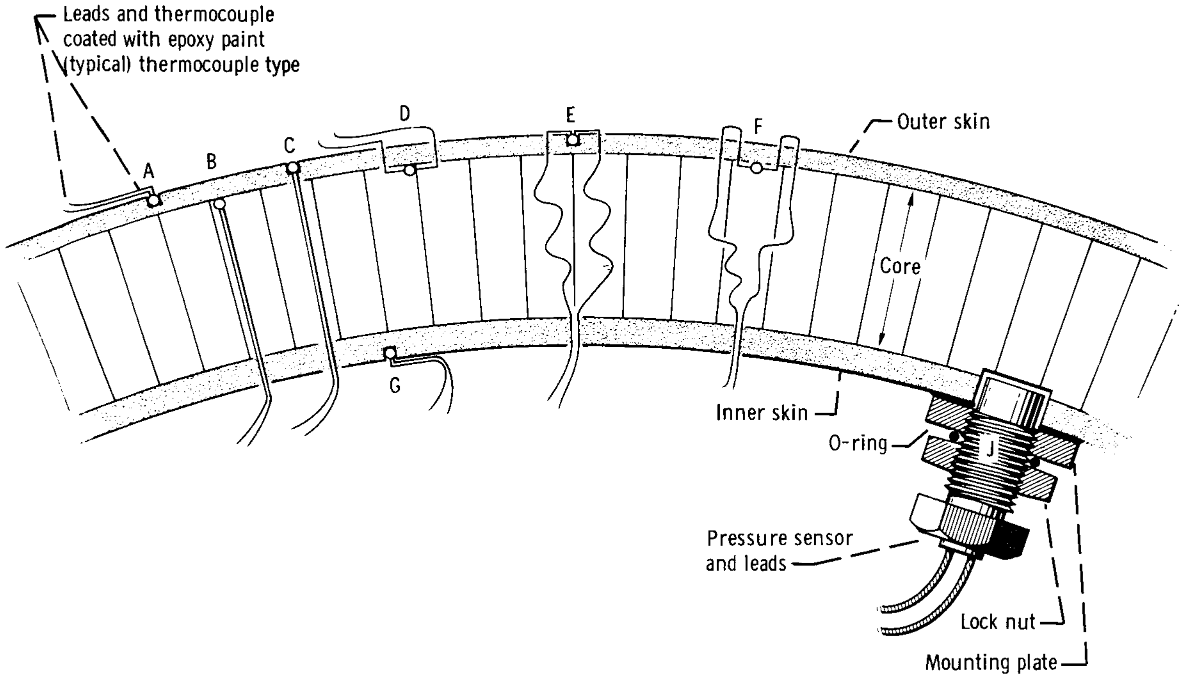
Test	Conditions				Results								
	Specimen			Simulated heating	Maximum temperature						Maximum core pressure buildup		Remarks
	Number	Size			°F			K			psia	N/cm^2	
		in.	cm		Outer skin	Bond line	Inner skin	Outer skin	Bond line	Inner skin			
1	1	24 by 24	60.9 by 60.9	Maximum design	835	Not sensed	125	719	Not sensed	325	Not sensed	Not sensed	Top layer of outer skin delaminated at center
2	2	24 by 24	60.9 by 60.9	Maximum design	865	[a]520	125	736	[a]544	325	17.2	11.85	Outer skin completely delaminated from core at T + 147 sec
3	3	24 by 24	60.9 by 60.9	AC-4, 3-sigma trajectory	755	670	150	674	627	339	17.0	11.7	Outer skin porous to gaseous nitrogen at 7 psi (4.82 N/cm^2) after test
4	[b]4	24 by 24	60.9 by 60.9	AC-4, 3-sigma trajectory with effect of 230 sublimate	230	200	Not sensed	383	367	Not sensed	14.0	9.65	--------------------
4A	[b]4	24 by 24	60.9 by 60.9	AC-4, 3-sigma trajectory with effect of 230 sublimate	225	200	Not sensed	380	367	Not sensed	[c]13.8	[c]9.52	Specimen pressurized from 13.8 psia (9.52 N/cm^2) at T + 160 sec to 50 psia (34.4 N/cm^2) at T + 200 sec; inner skin delaminated from core at T + 223 sec
5	5	7 by 7 with 0.04 in. 230 sublimate	17.8 by 17.8 with 0.1 cm 230 sublimate	AC-4, 3-sigma trajectory	[d]200	[d]180	Not sensed	[d]367	[d]355	Not sensed	[d]14.4	[d]9.93	Test not valid; excessive heating imposed on specimen
6	6	7 by 7 with 0.04 in. 230 sublimate	17.8 by 17.8 with 0.1 cm 230 sublimate	AC-4, 3-sigma trajectory	235	220	Not sensed	386	378	Not sensed	14.0	9.65	Approximately 25 percent of 230 sublimate sublimed

[a]Questionable data because of instrument installation failure.
[b]Skin to core delaminations: 2-in. (5.08-cm) diam flaw under outer skin; 2-in. (5.08-cm) diam flaw under inner skin.
[c]No core pressure buildup prior to pressurization from external source.
[d]Value at T + 130 sec prior to sublimation of all 230 sublimate.

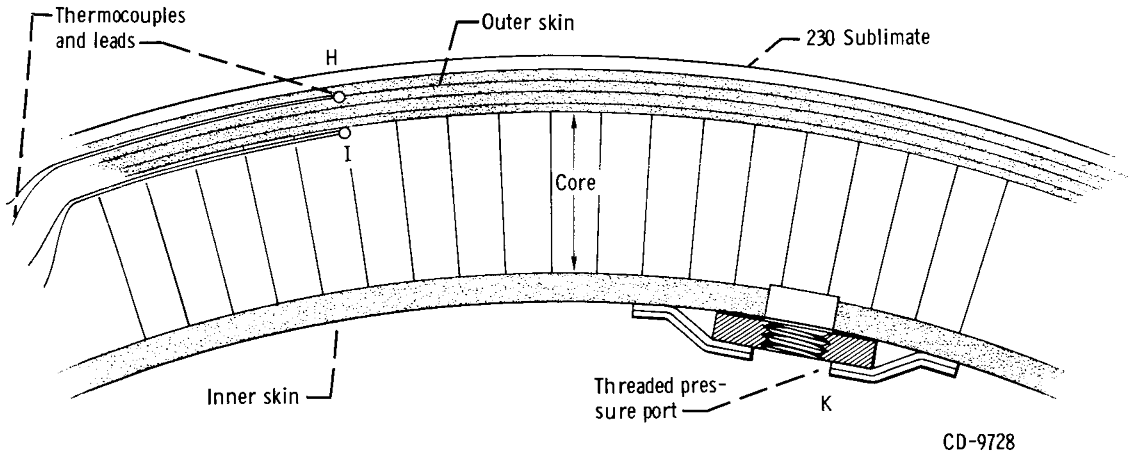


Station		Thickness of - Inner skin	Core	Outer skin
-46	(-117)			
-35	(89)	0.03 (0.08)	0.14 (0.36)	0.03 (0.08)
		.06 (.15)	.50 (1.27)	.06 (.15)
0	(0)			
		.04 (.10)	.80 (2.03)	.05 (.13)
20	(51)			
		.04 (.10)	.80 (2.03)	.04 (.10)
		.04 (.10)	1.50 (3.82)	.04 (.10)
142	(361)			
		.02 (.05)	.15 (.38)	.02 (.05)
151	(384)			
		.04 (.10)	1.50 (3.82)	.04 (.10)
219	(558)			

Figure 1. - Centaur-Surveyor nose fairing configuration. (All dimensions are in inches (cm).)



(a) Typical for 24-inch (60.9-cm) square specimen.



(b) Typical for 7-inch (17.8-cm) square specimen.

Figure 2. - Installation of thermocouples and pressure taps on nose fairing specimens for use in aerodynamic heating investigation.

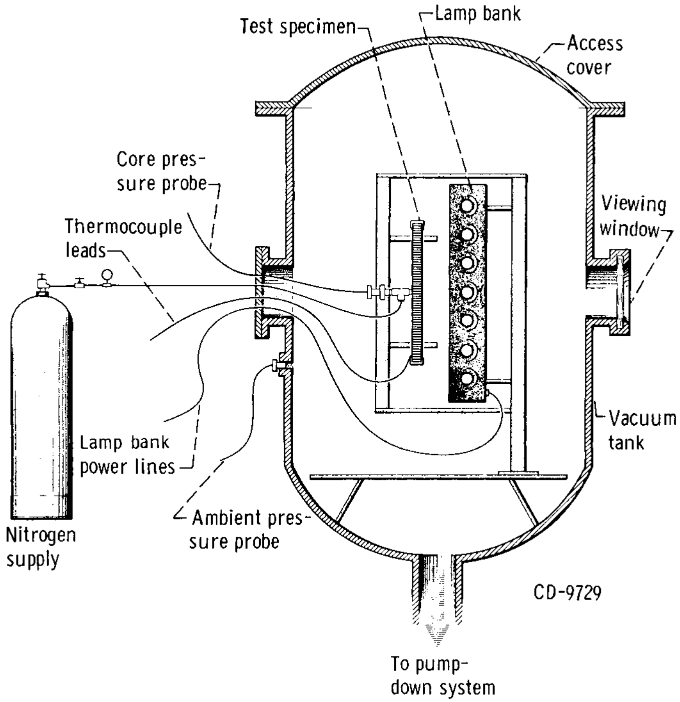


Figure 3. - Apparatus used in nose fairing aerodynamic heating investigation.

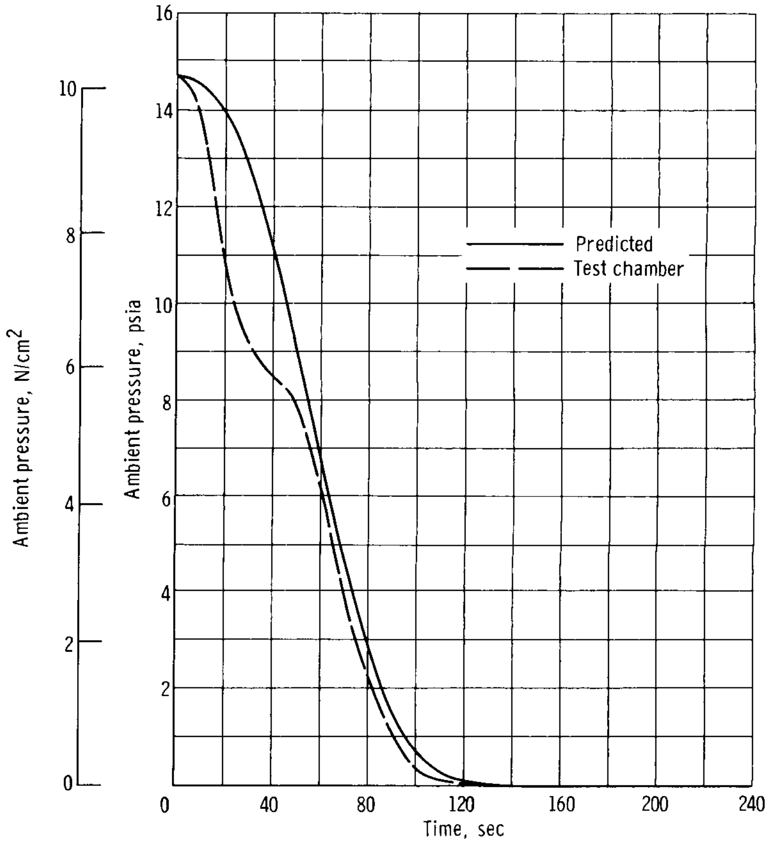
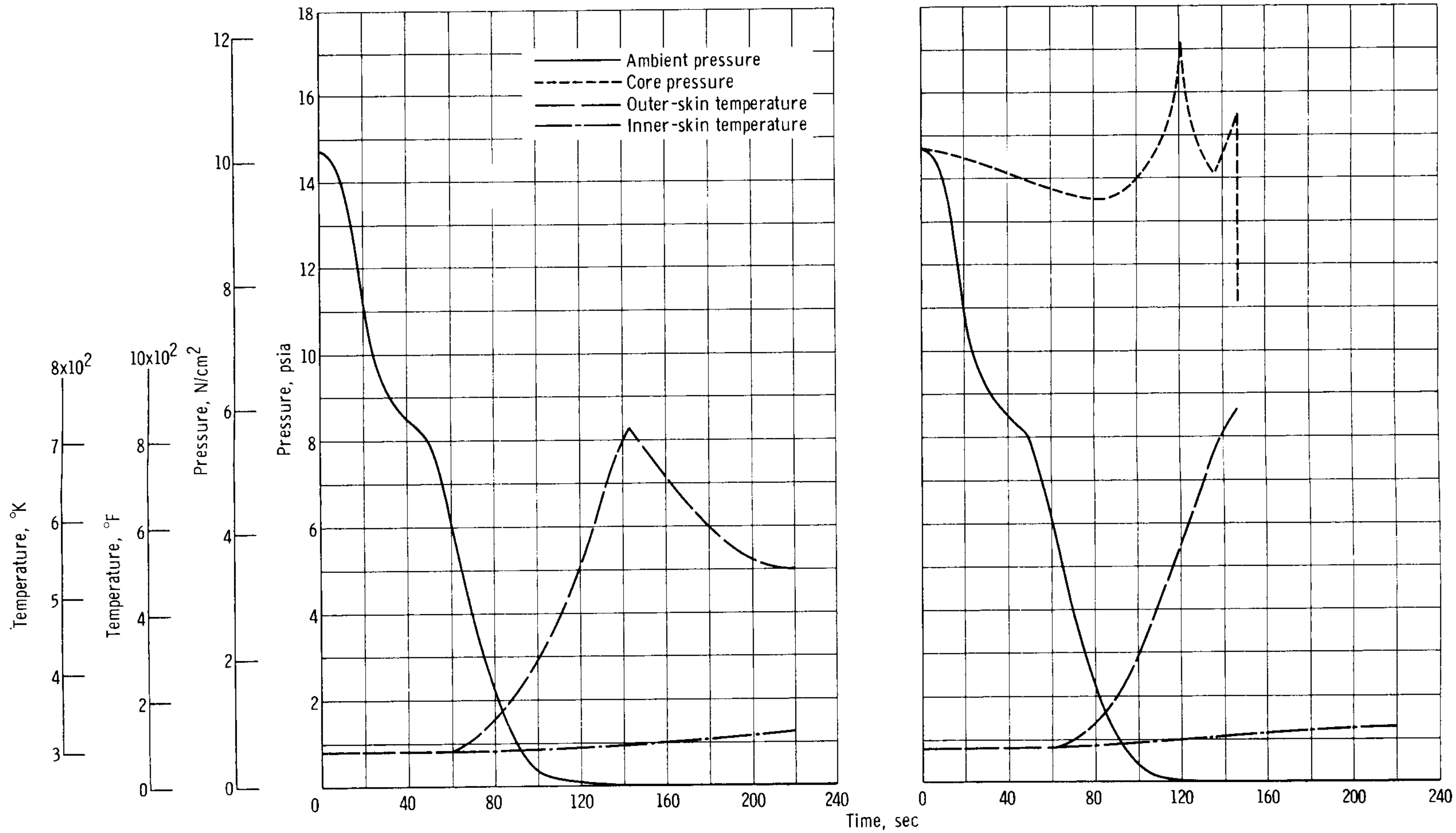


Figure 4. - Comparison of AC-4 predicted ambient flight pressure and test chamber pressure.



(a) Specimen 1.

(b) Specimen 2.

Figure 5. - Pressures and temperatures of nose fairing specimen for maximum design heating.

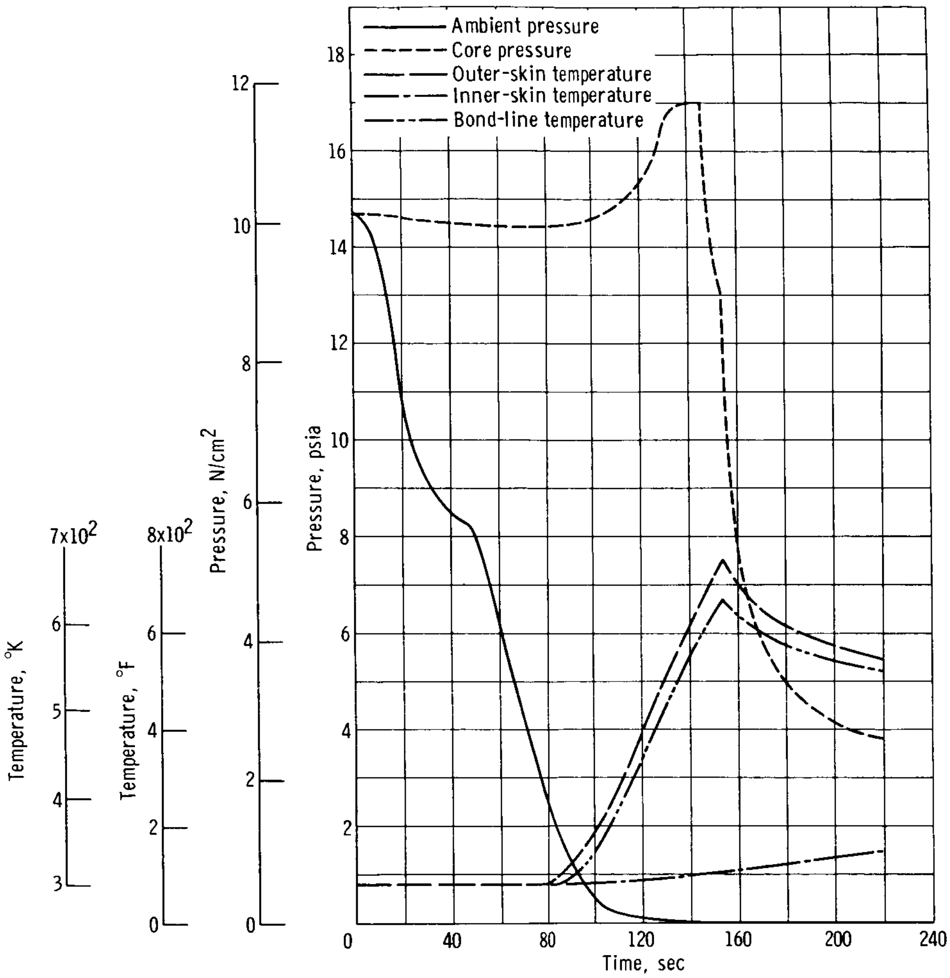
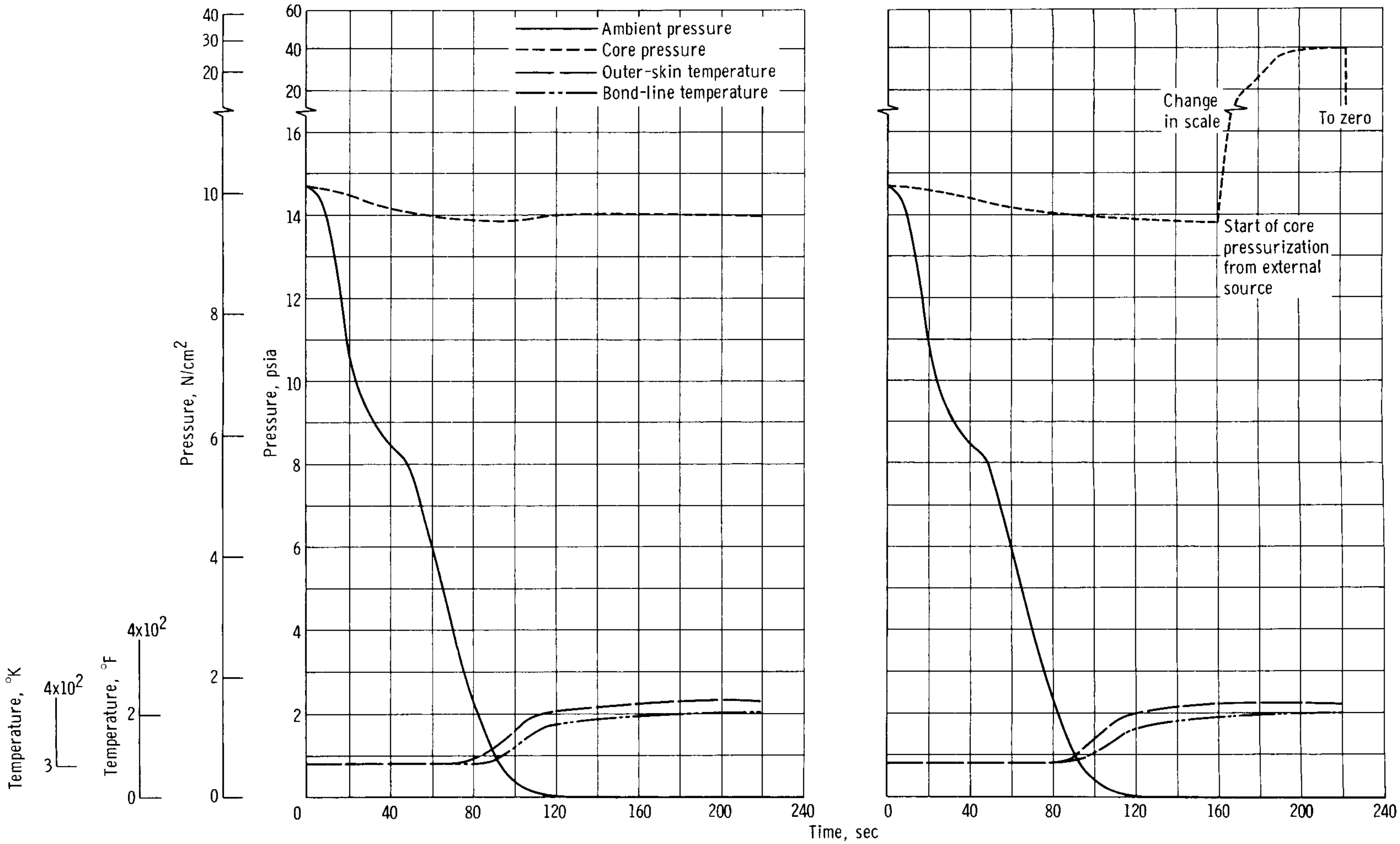


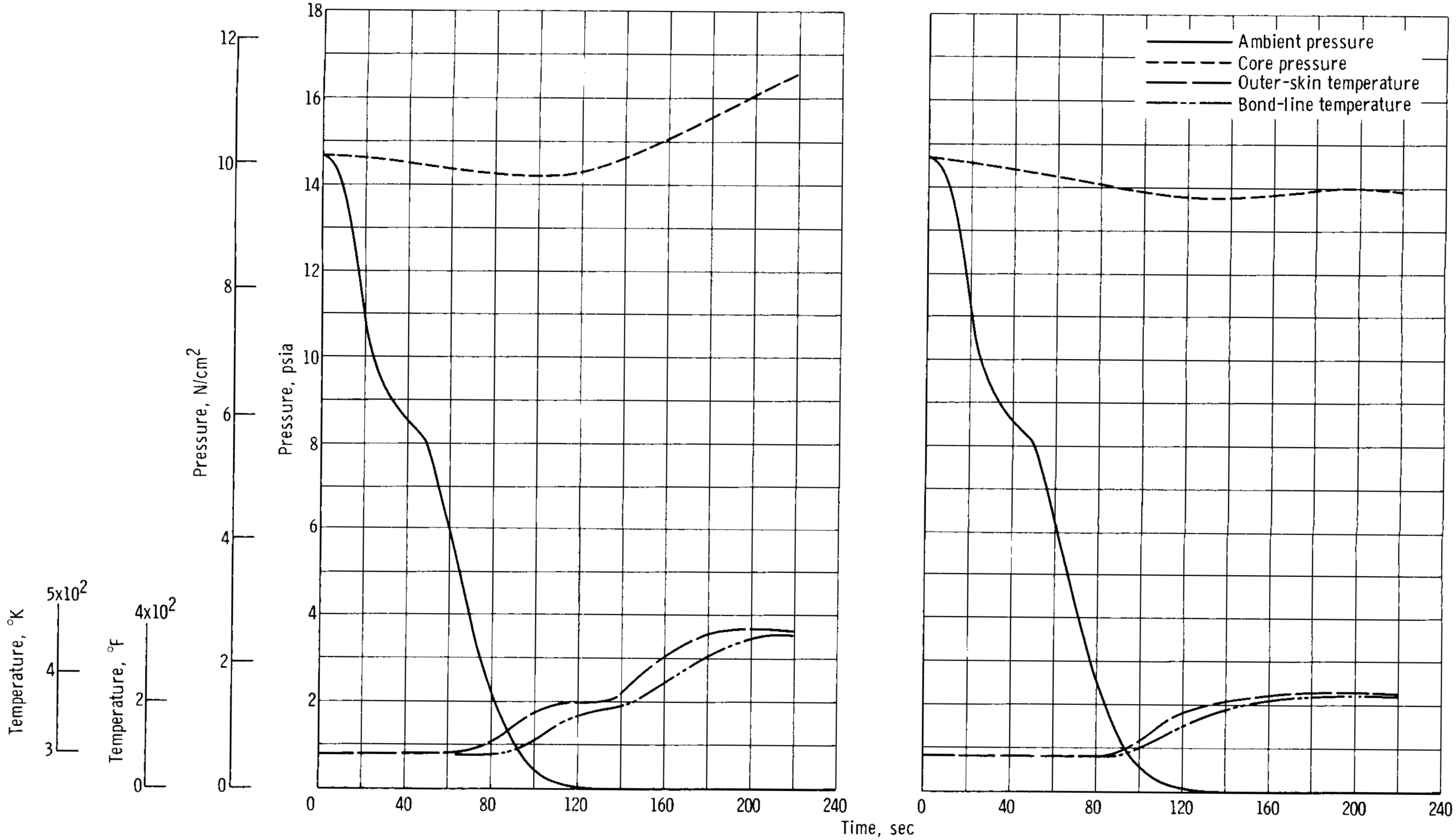
Figure 6. - Pressures and temperatures of nose fairing specimen 3 for 3-sigma low-trajectory heating.



(a) Specimen 4.

(b) Specimen 4 with core pressurization.

Figure 7. - Nose fairing specimen pressures and temperatures for heating of outer skin to 230 sublimate sublimation temperature.



(a) Specimen 5 with excessive heating and complete sublimation of 230 sublimate coating.

(b) Specimen 6 with 3-sigma low-trajectory heating.

Figure 8. - Pressures and temperatures of nose fairing specimen with 0.04 inch (0.1 cm) of 230 sublimate coating.

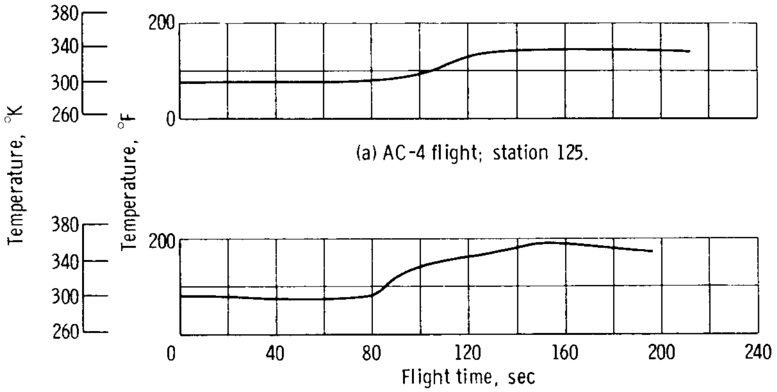


Figure 9. - Centaur nose fairing outer-skin flight temperatures (under 230 sublimate).

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